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SPACECRAFT

The present invention relates to a spacecraft and particularly to a spacecraft having a sun ray blocker device for shading thermal radiator surfaces on said spacecraft from solar heating.

The following patents are generally representative of the prior art in the broad fields of solar array related sun shields, solar array deployment mechanisms, and the thermal control of radiator surfaces for various types of spacecraft.

United States Patent 4,133,502 to Andrew Anchutin describes a plurality of arrays of solar cells which are symmetrically stored about a spacecraft during launch to provide symmetrical loading. When the spacecraft is in operational configuration, the solar arrays are deployed adjacent each other on one side of the spacecraft to effectively form a single array and the single array may be oriented to face the Sun by a common drive mechanism.

United States Patent 4,508,297 to Guy G. Mouilhayrat et al, describes an equatorial orbit satellite with solar panels having blades with a median line inclined at a certain angle relative to the equatorial plane. Thus, the field of vision of the antennas is free and disturbing torques become acceptable.

United States Patent Number 5,372,183 to Harold P. Strickberger describes a spacecraft adapted for operation

in a low inclination angle earth orbit which comprises north, south, east and west panels defining a spacecraft interior volume. The north and south panels are oppositely disposed with respect to each other and the east and west panels are oppositely disposed with respect to each other. The spacecraft interior volume generally and preferably lacks structural elements that substantially restrict thermal radiation among the panels. The north and south panels, to which spacecraft equipment is usually mounted, each include conductive heat pipes for reducing the temperature difference across each panel. The exterior surfaces of the north, south, east and west panels have a covering, preferably of optical solar reflectors (OSRs), for radiating thermal energy therefrom, wherein the OSRs have a solar absorptivity that is substantially less than their thermal emissivity. The interior surfaces of the north, south, east and west panels have a covering for effectively radiating thermal energy between and among the panels across the interior volume.

United States Patent 4,725,023 to Haruo Shiki describes a geostatic satellite which comprises a spinning drum for stabilization which spins around an axis of rotation which is parallel to the axis of the Earth. A paddle member loaded with solar cells is directly rotatable about the same axis and is controlled such that the solar cells face the Sun. A de-spun platform supports communication gear and maintains the gear pointed to a relatively fixed point on Earth. A shading device for shading the electronics laden de-spun platform from the Sun is attached to the paddle member

and rotatable therewith. Thereby, the shading device will always be disposed between the Sun and the de-spun platform. However, the shading device also blocks thermal radiation from the platform and also itself heats up in sunlight and radiates heat towards the platform, decreasing the efficiencies of heat transfer from the spacecraft to space.

European Patent Application 98401320.1 by Aerospatiale Societe Nationale Industrielle describes a satellite comprising a face used as a thermal radiator (i.e. a radiator-face) for equipment on board and lying in the path of solar radiance, a solar panel electric generator constantly oriented towards the Sun and projecting from the centre of the radiator-face, and a screen at the edge of the satellite fixed to the solar panel by a connecting arm and stopping the solar radiance directed towards the radiator-face. The screen reduces variations in temperatures of the radiator-face. Alone, however, coatings on the exterior of the screen are inadequate to prevent the screen from heating up in sunlight and radiating significant heat towards the radiator-face. Consequently, the effective radiation view factor of the radiator-face to deep space is significantly limited, the efficiencies of heat transfer from the radiator-face to space are limited, and temperatures of the radiator-face and the equipment on board are unduly high.

For discussions of radiation view factor and related factors see, for example, pp. 202-234, "Principles of

Heat Transfer", Frank Kreith, Second Edition, University of Colorado, 1965, and p. 426, "Principles of

Communication Satellites", Gary D. Gordon and Walter L. Morgan, John Wiley and Sons, Inc., 1993.

European Patent Application 87402281.7 by Centre Nationale d'Etudes Spatiale describes a device for a geostationary satellite comprising a screen fixed on a crown rotated by a motor-driven pinion so as to orient the screen towards the Sun and protect a radiator used to cool detectors of infrared instruments. The curvature and extent of the screen, however, significantly limit the effective radiation view factor of the radiator to deep space, raising the temperatures of the radiator and detectors unduly.

European Patent Application EP 91301447.8 by GEC-Marconi Limited describes a geostationary satellite comprising a pair of solar panels extending therefrom, the planes of the panels lying parallel to the axis of rotation of the satellite when in orbit. The solar panels are offset with regard to an axis of rotation of the satellite passing through the centre of mass of the satellite. Attached to each solar panel is a blanking plate, substantially co-planar with the solar panel and extending to a plane containing the face of the satellite from which the solar panel is supported. Optionally, side panels (also blanking plates) may also be attached along the north-south edges of the solar panels. The solar panels, blanking plates, and side plates provide masking of the Sun's rays for the faces of the satellite on which the solar panels are mounted. The masking of the Sun reduces the variations in temperatures of the shaded panels. The solar panels, however, are heated by the Sun and radiate heat towards the shaded panels,

decreasing the efficiencies of heat transfer from the shaded faces to space, and raising the average temperature of the shaded faces and the equipment on board. Furthermore, since solar arrays have relatively high mass, the offsetting of their centres of mass from their axes of rotation poses substantial difficulties, related to varying system mass-properties, in design of the satellite system.

United States Patent 5,527,001 to Teledesic Corporation describes a modular communication satellite comprising a solar array that completely shades the rest of the satellite from the Sun in operation in orbit. The masking of the Sun reduces the variations in temperatures of the rest of the satellite. The solar array, however, is heated by the Sun and radiates significant heat towards the rest of the satellite, decreasing the efficiencies of heat transfer from the rest of the spacecraft to space, and raising the average temperature of the rest of the spacecraft unduly.

Notwithstanding the prior art, the present invention is neither taught nor rendered obvious thereby.

It is an object of this invention substantially to reduce or eliminate the direct and indirect solar heating of certain spacecraft radiator-panels, and to also minimize the magnitude of any reduction in the radiation view factor of the (shielded) radiator panel to deep space. In order to achieve that objective, the materials and design selected for the sun ray blocker device, which will be discussed below, should ideally provide all of the following: minimum blockage of the field-of-view to

deep space of its associated radiator surface(s), low absorption of the solar energy incident on its front (sunward) surface, high radiation of absorbed thermal energy back to space, and high insulation of heat between the front (sunward) and back (anti-sunward) sides of the sun ray blocker device.

It is also desirable to provide a sun ray blocker device that is capable of greatly reducing or eliminating solar energy incident on those sides of certain

spacecraft relative to which the Sun direction makes a low angle. The types of spacecraft to which the present

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invention applies include some spacecraft for operation in equatorial or low inclination orbits, and in sun synchronous orbits with low orbit-sun angles. In the case of three-axis stabilized, Earth-pointing, geostationary spacecraft for example, these shaded sides are either or both of the north and south main-body panels. In the case of the sun synchronous spacecraft for example, the shaded sides are either or both of the sides or main-body panels that face out along the pitch axis (i.e. that face parallel to the orbit normal and anti-normal). The present invention can also be applied to types of spacecraft, other than geostationary and sun synchronous types, upon which the solar illumination is incident at low angles relative to thermal radiator surface. In those spacecraft it is those main thermal radiator surface that can be shaded by the present invention device.

According to a first embodiment of the invention there is provided a spacecraft for orbiting a sunlit celestial body, the spacecraft including a thermal radiator surface for radiating heat from the spacecraft into space, and a sun ray blocker device mounted on said spacecraft for shielding said thermal radiator surface from rays of sunlight, characterised in that said sun ray blocker device includes at least one sun blocker component, said sun blocker component being locatable, in an operational configuration, on a sun line from said thermal radiator surface and being of suitable shape, size, and orientation for placing in shadow up to the whole of said thermal radiator surface from sunlight, said sun blocker component having a surface intended to face the Sun in use and an opposed surface intended to

face away from the Sun in use, said sun blocker component being adapted for achieving a high radiation view factor from the thermal radiator surface to deep space by means including thermal insulation material located between the sun-facing surface and the opposed surface for restricting heat flow through said sun blocker component between said sun-facing surface and said opposed surface.

Conveniently, the sun-facing surface is thermally insulated from the opposed surface by multi-layer insulation (MLI).

A preferred inventive feature will be seen to reside in said sun blocker component being further adapted for achieving a high radiation view factor from the thermal radiator surface to deep space by means including a region of said opposed surface being adapted to lie, in an operational configuration, substantially in a plane for limiting a radiation view factor from said opposed surface to said opposed surface.

A preferred inventive feature will be seen to reside in said sun blocker component being further adapted for achieving a high radiation view factor from the thermal radiator surface to deep space by means including a region of said opposed surface being adapted to face, in an operational configuration, at an angle away from said thermal radiator surface for limiting reflection by said sun blocker component of thermal energy from said thermal radiator surface back to said thermal radiator surface.

A preferred inventive feature will be seen to reside in said sun blocker component being further adapted for achieving a high radiation view factor from the thermal radiator surface to deep space by means

including a dimension and/or a shape of said sun blocker component, in an operational configuration, serving to limit a corresponding geometric radiation view factor from said thermal radiator surface to deep space.

Preferably an effective radiation view factor for thermal radiation from the thermal radiator surface

(11,12,1804, 2121,2721) to deep space is significantly greater than a corresponding geometrical radiation view factor to deep space, and in particular, for a geostationary spacecraft where the geometrical radiation view factor to deep space is 0.65, the effective radiation view factor to deep space is at least 0.87.

Advantageously the sun-facing surface has a low solar energy absorptivity of less than 0.5.

Advantageously the sun-facing surface includes a solar cell panel for supplying electrical power to the spacecraft.

Preferably the sun-facing surface has a high thermal emissivity of higher than 0.7.

A preferred inventive feature will be seen to reside in the sun ray blocker device being adapted for a re-configuration involving movement between a stowed, non-operative position and a deployed, operative position after launch of the spacecraft.

Conveniently the sun ray blocker device includes an attachment arm for attaching the sun blocker component to the spacecraft.

Advantageously the attachment arm is attached by a hinge means to the sun blocker component and/or by a second hinge means to the spacecraft.

Advantageously the sun ray blocker device includes a motor for moving said sun ray blocker device between the stowed position and the deployed position.

Preferably locating means are provided for locating the sun ray blocker device with respect to the thermal radiator surface which include adjustment means to maintain up to the whole of the thermal radiator surface in shadow irrespective of changes in the attitude and/or orbital position and/or orbit of the spacecraft.

A preferred inventive feature will be seen to reside in the adjustment means including a variable length attachment arm for attachment of the sun blocker component to the spacecraft.

Advantageously the attachment arm is a scissors arm.

Alternatively the attachment arm is formed of articulated portions which may be mutually articulated during rotation to vary an effective length of the attachment arm.

A preferred inventive feature will be seen to reside in the adjustment means including carriage means for carrying the sun blocker component and transport means for moving the carriage with respect to the spacecraft.

Conveniently the transport means includes guide means and the carriage means includes drive means to drive the carriage along the guide means.

Preferably the transport means includes guide means and motive means that are external to and connected to the carriage means , the external motive means being driven by

drive means to move the carriage means along the guide means.

Alternatively the transport means includes an annulus rotatable in a circular path defined by bearing means, the annulus being driveable by drive means to move the carriage along the path defined by the bearing means.

Conveniently the spacecraft has a solar cell array adapted for tracking movements of the sun relative to the spacecraft, wherein the adjustment of the location of the sun ray blocker device in relation to the thermal radiator surface is synchronised with the tracking movement of the solar cell array, when in normal operation.

Conveniently the sun ray blocker device is mounted on the solar cell array or on means carrying said solar cell array.

Advantageously the solar cell array is adapted for tracking the movement of the sun by rotation of the solar cell array about an axis of rotation of the solar cell array such that the sun blocker component also rotates about said axis of rotation of the solar cell array.

Conveniently the thermal radiator surface is orthogonal to the axis of rotation of the solar cell array so that the sun blocker component rotates about an axis normal to the thermal radiator surface.

Conveniently adjustment means for attachment of the sun blocker component to a solar cell array assembly are such that a distance between the sun blocker component and the solar cell array assembly may be varied during rotation of the sun blocker component.

Conveniently the sun ray blocker device is adapted for tracking the movement of the sun by rotation of the sun ray blocker device about an axis of rotation of the sun blocker device which is orthogonal to the thermal radiator surface so that the sun blocker component rotates about an axis normal to said thermal radiator surface.

Conveniently means are provided for adjusting the form and/or size of the sun blocker component.

Conveniently the spacecraft includes control means for controlling the spacecraft so as to maintain an angle between a sun line and the thermal radiator surface below a predetermined angle by adjustment of the orbit and/or attitude of the spacecraft in use.

Preferably the predetermined angle is 60 degrees.

More preferably the predetermined angle is 45 degrees.

Most preferably the predetermined angle is 23.5 degrees.

Advantageously the control means is adapted to maintain the thermal radiator surface substantially parallel to a plane of an orbit of the spacecraft.

Alternatively the control means is adapted to maintain the spacecraft in a sun synchronous orbit.

Alternatively, the control means is adapted to maintain the spacecraft in an equatorial or low-inclination orbit.

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~~said solar cell panel extending outwardly from said spacecraft, the improvement which comprises:~~

~~attaching at least one sun ray blocker device to said at least one solar cell panel, said at least one device being either a north blocker device or being a south blocker device and corresponding to said at least one solar cell panel, each of said at least one sun ray blocker device being positioned forwardly from and offset relative to a solar cell surface of a solar cell panel and at a predetermined angle to either of said north panel and said south panel, said north panel or said south panel, said sun ray blocker device being positioned so as to cast a shadow on at least a majority of the exposed surface of its corresponding north or south panel during solar exposure thereto.~~

~~According to another embodiment of the invention there is provided in a three axis stabilised low inclination orbit spacecraft for orbiting about the earth and having two sets of solar cell array assemblies having solar cell arrays, one set being a north solar array assembly and the other being a south solar array assembly, said assemblies each being mounted on an axle so as to be controllably rotated from said spacecraft about an axis of rotation so as to face the sun, said spacecraft having an earth panel which is generally pointing to the centre of the earth, an opposite panel known as a zenith panel, which faces away from the centre of the earth and sharing the same planar normal vector as said earth panel, an east panel and a west panel, said east panel and said west panel having their planar normal vector laying on an orbital plane pointing to the~~

The present invention preferably provides a sun-synchronous sun ray blocker device (not to be confused with sun synchronous orbits referred to elsewhere herein) for use in a spacecraft designed to orbit around a planet with solar incidence at low angles to their thermal radiator surfaces, i.e. with sun directions close to the planes of the individual radiator surfaces. Preferred embodiments of the present invention are spacecraft for operating in an orbit plane oriented at a

low angle (or within a range of low angles) to the sun direction, the said spacecraft having a thermal radiator surface that is oriented approximately parallel to the orbit plane and a solar array assembly that is rotated about an axis approximately perpendicular to the orbit plane nominally at the orbital rate. Examples of appropriate orbits are: (a) low inclination orbits around the Earth (including nominally equatorial orbits), and (b) sun synchronous orbits with low orbit-sun angles (which around Earth and Mars, for example, are nominally polar orbits). The term "spacecraft" as used herein includes satellites and other space bound vehicles.

Mounted on any spacecraft to which the present invention is applied is at least one device for blocking sun rays and thereby preventing them from directly impinging on a radiator surface of the spacecraft.

In many embodiments of the present invention the individual spacecraft will have at least one solar array assembly (comprising solar cell panels and rotary axial booms) which may be used as mounting support for the sun ray blocker device(s), so that the combination assembly of solar array assembly and sun blocker device(s) is operationally controllably rotated together as an integral unit to track the Sun throughout the orbital revolutions of the spacecraft, said solar array assembly being mounted on the spacecraft so that operationally in orbit it can be rotated about an axis that is maintained oriented approximately perpendicular to the orbit plane in a manner such that the solar-cell side of the solar cell panels is maintained sun facing and substantially

perpendicular to the Sun direction. Because then the sun ray blocker device may rotate integrally with the solar array assembly, it is able to prevent sun rays from directly impinging on all or part of an associated thermal radiator surface(s), whose plane is maintained approximately parallel to the orbit plane, thus creating a continuous steady and benign thermal environment for the thermal radiator surface.

Spacecraft operated in orbits with low orbit-Sun angles (the angle between the orbit plane and the Sun) are prime candidates for application of the current invention device. Various different frequently-utilized types of orbits feature low orbit-Sun angles. Currently, among the most utilized types of orbits with low orbit-Sun angles are (a) low inclination and nominally-equatorial orbits, including geosynchronous orbits, and (b) the subset of sun synchronous orbits with low orbit-Sun angles. Sun synchronous orbits maintain a little-varying orbit-Sun angle as the planet revolves around the Sun. The Earth revolves around the Sun once per year.

One type of spacecraft operated in a nominally equatorial orbit around a planet, e.g. the Earth, or in particular, a geosynchronous orbit, is frequently used for the purposes of telecommunications, broadcasting, monitoring ecological conditions, global positioning, remote sensing, surveillance and weather forecasting.

Another type of satellite operated in nominally sun synchronous orbits around planets, e.g. the Earth, with low orbit-Sun angles, is frequently used for the purposes

of weather monitoring and remote sensing of the planet and its atmosphere. Some of the benefits of these sun synchronous orbits are: low spacecraft altitudes, frequent over-flight of the planet within close proximity of virtually all latitudes and longitudes, and near constant angle of solar illumination on the day side of the orbit.

Means of adjusting the attitude and orbit of spacecraft are well known, for example, are described in "Principles of Communications Satellites" by Gary D Gordan and Walter L Morgan published by John Wiley & Sons 1993, pages 12-14, 55-58 and in "Spacecraft Attitudes, Termination and Control" by James R Wertz published by Kluwer Academic Publishers 1978. Attitude and orbit control may for example be provided by the use of thrusters and/or momentum or reaction wheels.

Typically, the attitude (i.e. the orientation) of these types of satellite is controlled so that as the satellite orbits the planet part of its payload equipment steadily faces approximately toward the center of the planet, while the solar arrays are maintained sun pointing. The attitude (orientation) control systems of such spacecraft belong to various classifications that are well known within the space industry. For example, two of the more currently prominent types of attitude control system are commonly referred to as "three-axis-stabilized" control systems and "spin stabilized" control systems. The present invention device functions independently of the type of attitude control system, and independently of the orientation of spacecraft equipment

other than the orientation of the thermal radiator surfaces that the device shields from solar energy. In these types of spacecraft, the performance of the present invention device is generally better the closer the shadowed thermal radiator surface is to being parallel to the orbit plane (which in these types of spacecraft is maintainable at a low angle to the sun line).

Hereinafter, the concept of a "model spacecraft" is defined and employed in order to avoid the distraction of multiple lengthy descriptions of diverse spacecraft to which the present invention device may be applied. The model spacecraft is used herein, somewhat like a tailor's dummy, in order to facilitate the illustration and explanation of features, functions, and examples of applications of the present invention device.

By definition the model spacecraft has a basic, deployed (i.e. unfolded), structural configuration that is typical of many current three axis stabilized satellites, and a corresponding operational mode that is typical of a 3-axis-stabilised geostationary Earth pointing spacecraft. Note that this definition was selected on the basis of current estimates of the most frequent future application of the present invention device. The definition of the model spacecraft could equally well have been based on typical characteristics of another relevant type of spacecraft, for example an Earth-pointing spacecraft in a sun synchronous orbit with a low orbit-Sun angle.

Referring to FIGURES 1 and 5, the basic structural configuration of the model spacecraft is based on a main body in the form of a hollow, right parallelepiped. For

the stated purposes of using the concept of the "model spacecraft" herein, it is useful to consider the main body as comprising six principal, planar, structural panels. The external surfaces of one opposing pair of the six panels that form the main body of the model spacecraft constitute the mounting sites for the thermal radiator surfaces that are shielded from direct solar heating by means of the present invention device. Mounted on one or each of these two radiator-bearing panels, and extending perpendicularly outwards therefrom, is a solar array assembly, comprising a rotary solar array boom to which are attached solar cell panels.

That is not to say that application of the present invention device is limited to spacecraft with a structural configuration and/or an operational mode resembling that(those) of the model spacecraft. For example, the present invention device is also applicable to: sun synchronous spacecraft in orbits with low orbit-Sun angles; spin stabilized spacecraft; spacecraft with polyhedral and/or irregular structures; spacecraft that are not nadir pointing; spacecraft that are not geostationary; spacecraft with solar arrays that deploy and subsequently lie along axes that are not perpendicular to the radiator-bearing panels; etc.

Much of the text herein that supports the accompanying claims is written with reference to the model spacecraft or to 3-axis-stabilised Earth-pointing geostationary spacecraft. Regardless, the supporting text also applies to applications of the present invention device to other suitable types of spacecraft. For example, the principal relevant difference between

many suitable sun synchronous spacecraft (for polar orbits at Earth and

Mars at least, where the polar axes lie close to the planes of sun-synchronous orbits around them) and the model spacecraft is that the plane of the thermal-radiator surface(s) that is shaded by the present invention device is approximately parallel to the axis of rotation of the planet (rather than perpendicular to the axis of rotation of the planet as for geostationary spacecraft like the model spacecraft). Accordingly, the supporting text describing spacecraft like the model spacecraft is easily read as it relates to these suitable sun synchronous spacecraft, for example by substituting in "pitch axis panel" or "orbit normal panel" to replace "north/south panel", and substituting in "velocity panel" or "roll axis panel" to replace "east/west panel".

In order to provide functional services in an operational orbit, the model spacecraft has one of the six structural panels of its main body continuously facing the planet, e.g. the Earth. That panel is referred to as the earth panel or the nadir panel. A vector that is outward-from and normal-to the earth panel is parallel to the (body fixed) yaw axis of the spacecraft. In the model spacecraft the yaw axis is maintained nominally parallel to the nadir direction, i.e. is nominally pointed toward the center of the planet. Because the model spacecraft operates in a geosynchronous orbit, which is nominally circular, the yaw axis of the model spacecraft is maintained nominally perpendicular to the velocity vector of the spacecraft. A vector that is outward-from and normal-to the plane of the structural panel opposite the nadir panel is parallel to the negative yaw axis. That main-body panel of three

axis stabilised Earth pointing geostationary spacecraft like the model spacecraft is usually referred to as the zenith panel or anti-earth panel.

Another opposing pair out of the six structural panels comprising the main body of the model spacecraft are oriented so that, nominally or approximately, vectors that are outward-from and normal-to their planes lie in the orbital plane and are perpendicular to the yaw axis and to the nadir and zenith directions. These outward normal vectors are parallel and anti-parallel to the positive and negative (body fixed) roll axes of the spacecraft. Because the geosynchronous orbit of the model spacecraft is circular, the roll axes of the model spacecraft nominally coincide with the velocity and anti-velocity vectors of the orbital motion. For geostationary spacecraft the velocity of the spacecraft is eastward; and consequently these two main-body panels of spacecraft like the model spacecraft are generally referred to as the east and west panels.

Accordingly, the remaining two structural panels comprising the main body of the model spacecraft are oriented so that their planes are nominally or approximately parallel to the orbit plane. Vectors that are outward-from and normal-to the planes of these panels are parallel to the positive and negative (body fixed) pitch axes of the spacecraft. Since the geosynchronous orbit of the model spacecraft is nominally equatorial, the pitch axes of the model spacecraft are approximately parallel and anti-parallel to the spin axis of the Earth; and accordingly these two main-body panels of spacecraft

like the model spacecraft are referred to as the north and south panels.

To avoid unnecessary further repetition, in illustrating and explaining the features and functions of the present invention device herein, reference shall be made to application of the present invention device to the (previously defined) model satellite, which is operated in a three-axis-stabilized, Earth-pointing, geosynchronous mode.

Through each orbital revolution of a spacecraft like the model spacecraft, which in a preferred embodiment is around the Earth, the Sun sequentially directly illuminates the east, zenith, west, and nadir main-body panels. While illuminated (or insolated) thus these main-body panels absorb incident solar energy and their temperatures increase, which significantly reduces their net radiative cooling capability. If not countered by some means this can significantly limit the quantity of equipment (which dissipate heat into the spacecraft) that can be carried on board, and/or can result in undesirably elevated temperatures of associated spacecraft equipment. The north and south panels, however, generally face deep space during the entire orbit and only directly receive solar illumination and solar energy at relatively low

incidence angles on a seasonal basis. Because the direct input of solar energy into the north and south panels is relatively low to zero, these panels are the principal sites on spacecraft like the model spacecraft for the locations of thermal-energy radiator surfaces. The north panel is directly heated by the Sun for a duration of about 6 months (from about March 21st to about September 21st) at an incidence angle, defined as the angle between the panel plane and the sun vector, which seasonally increases from 0 degrees (when the sun vector is edge-on to the panel) to about 23.5 degrees followed by a decrease to 0 degrees again while the Sun is on the north side of the earth equator, i.e. during the northern spring and summer. The south panel is directly heated by the Sun for the remainder of the year, i.e. during the southern spring and summer, in a similar fashion and concomitantly with the north panel. These relatively low solar incidence angles favor use of the north and south panels for locating the principal thermal radiator surfaces of the spacecraft. At a maximum incidence angle of 23.5 degrees for the solar vector relative to the north and south panels the incident solar energy is approximately 40% of that for normal (perpendicular) incidence.

In the prior art numerous design practices have been employed to the surface treatment of the north and south panels in an effort to reduce the absorbed solar energy, thereby allowing more internal heat dissipation without raising the operational temperature level of the equipment that is thermally coupled to the panels. One example, optical surface reflectors (OSRs), which have a

high ratio of thermal emissivity versus solar absorptivity, have been widely used as the surface treatment of spacecraft thermal radiators. However, the seasonal solar heating of the spacecraft through OSRs still constitutes a significant amount of heat input to the spacecraft, which forces the spacecraft designer to lower the level of internal power dissipation to maintain an acceptable operating temperature for the spacecraft equipments. Solar energy absorbed by a spacecraft like the model spacecraft through its north and south panels has two obvious undesirable impacts on the performance of the spacecraft.

(1) It reduces the allowable level of internal power dissipation, which directly relates to the "value" of a spacecraft. The revenue from a spacecraft, especially a commercial communications spacecraft, is fundamentally limited by its capacity for power dissipation. A reduced allowable power dissipation level directly results in lower potential for revenue generation, which reduces the value of the spacecraft.

(2) The operating temperatures of the internal equipment are increased, and as a result the reliability of those components may be reduced. The reliability also relates to the life of a spacecraft, which directly relates to its "value" as well.

If the undesired solar heating were to be reduced, higher operational payload power would be allowable within the spacecraft and/or lower operating temperatures of the spacecraft equipments would be achieved.

Therefore, by virtue of the present invention, the spacecraft could be operated at a higher efficiency, with higher reliability, and would thereby generate revenue at a faster rate, all of which improvements would increase its value.

There is another important factor that affects the capability of a thermal radiator surface to reject heat to deep space: the "effective" radiation view factor (ranging from 0 to 1) from that panel to deep space. The ideal radiation view factor enabling a panel to reject maximum heat into deep space is unity (1). A device or means situated between the radiator surface and deep space could block the radiator's view to deep space and thus reduce the heat-radiating capability of the radiator.

The sun ray blocker device of this invention is mounted on the spacecraft, for example conveniently attached to the solar array assembly/assemblies of the spacecraft and rotating therewith. Since the primary function of the sun ray blocker device used in this invention is to provide a significantly more benign thermal environment for the principal thermal radiator surfaces (or panels), basically by shading them, the spacecraft should have at least one such surface. In the case of three-axis-stabilized Earth-pointing geostationary spacecraft, for example, there are two principal thermal radiator surfaces - the north and south panels; and accordingly at least two separate sun ray blocker devices can be included, one to shade each of these panels. Thus, the sun ray blocker device in the

present invention follows the movement of the Sun with respect to the thermal radiator panel(s) that it shades. In the case of a three-axis-stabilized Earth-pointing geostationary spacecraft, like the model spacecraft for example, the sun ray blocker device casts its shadow onto its associated thermal radiator surface, which is on either a north or a south panel, seasonally - through the six month long northern spring and summer in the case of the north panel, and through the six month long southern spring and summer in the case of the south panel. The (counter-productive) reduction in the radiation-view-factor of the thermal radiator surface caused by the presence of the associated present invention device is small; and the net effect of this reduction combined with the (beneficial) shading of the panel is a great improvement in the radiative efficiency of the radiator surface.

In addition to the foregoing, some of the considerations, advantages and parameters for the present invention device are as follows (others will become self-evident from the subsequent discussion of the FIGURES):

Variety in the operational form and size of the sun ray blocker device is permissible. The sun blocker component follows the movement of the Sun with respect to the spacecraft, and it blocks the Sun's rays by casting a shadow onto its associated thermal radiator surface at appropriate times, and it produces close to the minimum reduction in the effective radiation view factor to deep space of the thermal radiator that it shields, and it

satisfies other system requirements of the spacecraft (for example clear field of view requirements), as appropriate.

The material and/or the construction of the sun blocker component of the sun ray blocker device is preferably highly thermally insulating between its sun and anti-sun sides in order to provide the greatest practical effective radiation view factor and radiative efficiency of the radiator-surface shielded by the sun blocker component.

In its fully deployed configuration the sun blocker component may be mounted through a wide range of orientations relative to the radiator surface that it shields (for example, the angle 501 in FIGURE 12a below does not have to be 90 degrees i.e. a right angle. A requirement is that the sun blocker component casts shadow providing adequate coverage of the associated thermal radiator surface(s) on the spacecraft.

The ideal width of the sun blocker component is greater than either the width or the length of the radiator surface that it shields. However, the dimensions of the sun blocker component may be limited by other constraints. For example, in the launch configuration the dimensions of the sun blocker component may be limited by launch-envelope constraints, i.e. the size of the volume allowed for the spacecraft by the launch vehicle during launch. Therefore, it may be necessary to make the sun ray blocker component deployable to enable it to be folded or retracted for

launch and deployed in orbit. This can be achieved by hinged

deployment, slide extension, pre-offset or any other means to increase the width of the sun blocker

component (see FIGURES 16a, b and c, and 17a, b and c discussed below).

The mechanisms for extending, deploying, and supporting the sun blocker component may include various techniques and devices that are well known in the current state of the art of the design of mechanisms for spacecraft. For example the techniques and devices employed could include mechanisms constructed from well known device types such as: hinges, flaps, slides, spring motors, wax motors, detentes, cable/bolt cutters, split nut releases, pin pullers, hook and pin releases, etc. Alternatively or additionally, so-called "active" devices such as electrical motors may be used at the discretion of the spacecraft designer. For example, one or more electrical motor (for example a stepper drive motor) could be employed to produce the motions resulting in extension (and possibly also retraction) of the sun blocker component. Such active control could be utilized to facilitate certain operations of the spacecraft, for example station-keeping and attitude control operations for which displacing the sun blocker device from the exhaust plume fields of rocket thrusters would be beneficial.

The present invention device is applicable to spacecraft other than those spacecraft, like the model spacecraft for example, which operate in the low-inclination or equatorial orbits that have been described thus far herein. It is applicable to the broad class of spacecraft for which the solar illumination (insolation) is incident at low angles relative to the planes of the

surface(s) of their thermal-radiator surface(s).

A certain subset of spacecraft belonging to the set of spacecraft that are well known in the space industry as "sun synchronous" fulfil this requirement for low solar incidence angles on at least one thermal radiator surface; and the present invention is applicable to them. Within this subset of sun synchronous spacecraft is an even smaller but well known subset comprising those spacecraft that operate in orbits with low orbit-Sun angles and in which the thermal radiator surfaces are utilized while oriented close to parallel to the orbit plane. A sun ray blocker device according to this invention is applicable to those spacecraft, to provide them with a shaded, benign, and desirable thermal environment for their thermal-radiator surfaces basically by protecting them against direct solar heating. Note, however, that when the angle of incidence of direct sunlight on the thermal radiator surface is zero (i.e. for grazing incidence) or less the sun ray blocker device is unnecessary.

Heretofore the structural configuration and orientation of spacecraft to which the current invention device is applicable have mainly been described with reference to three-axis-stabilized Earth-pointing spacecraft for operating in low inclination or equatorial orbits, like the model spacecraft for example. The fundamental difference between those preceding descriptions and the structural configurations and orientations of the sun synchronous spacecraft to which the current invention device is applicable stems from the orientation of the orbit with respect to the axis of rotation of the planet. Within the space industry, sun

synchronous orbits are widely referred to as being "polar", since the orbit plane of a sun synchronous orbit, around Earth and Mars at least, lies within several degrees of the axis of rotation of the planet; and therefore nominally includes the planetary poles. Therefore, for the aforementioned particular subset of sun synchronous spacecraft to which the current invention device is applicable, the panels and the radiator surfaces on them that are thermally protected by the current invention device are generally not, strictly speaking, "north" and "south" panels. However, herein the terms "north" and "south" are occasionally used for convenience to indicate the panels that are thermally protected by the sun blocker device on spacecraft in sun synchronous orbits as well as on spacecraft like the model spacecraft, for example, in (nominally) equatorial orbits. The rationale is that in the particular, suitable, well known, and currently populous, aforementioned, subset of sun synchronous spacecraft the planes of thermal radiator panels shielded by the present invention device are also approximately perpendicular to the axis of the orbit (as for spacecraft like the model spacecraft in its orbital configuration and orientation). For both these types of spacecraft we could instead meaningfully refer to the protected panels and radiator surfaces as "pitch-axis" or "orbit normal" panels and surfaces, because the pitch axis of the spacecraft (which is parallel to the orbit normal) is nominally/approximately perpendicular to them and thereby defines their orientation.

Depending upon the requirements of the propulsion subsystem and/or the attitude control subsystem of the spacecraft, the spacecraft designer may elect to provide only one sun ray blocker device, i.e. on only one of the two sides of the spacecraft that face approximately along the pitch axis (e.g. on the north or the south panel for the model spacecraft). In any particular application there may be a preference for one side of the spacecraft over the opposite side because of other system requirements. For example, in a potential embodiment of the present invention device on a particular current design of geostationary spacecraft, the south side is preferred because of field of view requirements for attitude-control thrusters on the north side.

Again, if the spacecraft designer elects to do so, solar cells can be mounted onto the external surfaces of the sun blocker component to provide additional power to the spacecraft.

BRIEF DESCRIPTION OF THE DRAWINGS

The present invention should be more fully understood when the specification herein is taken in conjunction with the drawings appended hereto showing exemplary embodiments of the invention wherein:

FIGURE 1 is a simplified perspective view of a prior art three axis stabilized Earth-pointing geosynchronous spacecraft;

FIGURE 2 shows an east-panel based view of the prior art spacecraft illustrated in FIGURE 1 orbiting in a low inclination or an equatorial orbit;

FIGURE 3a shows a north-panel based, top view of the prior art spacecraft illustrated in FIGURE 1 orbiting about the Earth at different times of the day, and FIGURE 3b illustrates orbit-plane based views of that spacecraft at its noon, 6 a.m., and midnight positions and also establishes sun angles for different seasons of the year;

FIGURES 4a and 4b show the variation in the solar incidence angle on the north and south panels, respectively, of the prior art spacecraft illustrated in FIGURE 1 orbiting Earth, through one calendar year;

FIGURE 5 illustrates a perspective view of a spacecraft configuration according to the present invention, based on the prior art spacecraft illustrated in FIGURE 1;

FIGURES 6a, 6b and 6c illustrate top views of a present invention arrangement as applied to the prior art spacecraft illustrated in FIGURE 1. The views shown are simultaneously parallel to both the orbit-plane and the plane of the solar cell panels and the sun blocker components of the sun ray blocker device. Hereinafter this view direction is also referred to as "top view". FIGURES 6a, 6b and 6c show that as the spacecraft revolves around the orbit the earth panel always faces the Earth, and the cell-side of the solar array panels together with the

front (sunward) sides of the sun blocker components of the sun ray blocker devices always face the Sun;

FIGURES 7, 8 and 9 illustrate top views of present invention devices utilizing different attachment arrangements;

FIGURES 10a, 10b and 10c illustrate portions of top views of one of the solar array assemblies of a prior art spacecraft before, during, and after its deployment;

FIGURES 11a, 11b, 11c, 12a, and 12b show, in top view, aspects of the deployment and the function of present invention devices as applied to the prior art spacecraft shown in FIGURES 10a, 10b, and 10c;

FIGURES 13 and 14a show partial top views of two alternative present invention devices; and FIGURE 14b shows a partial back (anti-sun) side view of the arrangement shown in FIGURE 14a;

FIGURES 15a and 15b show partial top views of an alternative present invention device in its fully deployed and partially deployed configurations, respectively;

FIGURES 16a, b, and c show a view of an alternative present-invention device from the front (sunward) direction with the sun blocker device fully deployed, and two views from a direction orthogonal to both the front view and the (previously defined) top view directions

with the sun blocker device folded and deployed, respectively;

FIGURES 17a, b, and c show a different alternative present-invention device in the same views and deployed states as those shown in FIGURES 16a, b, and c;

FIGURE 18 shows a further embodiment of the invention;

FIGURE 19 shows another embodiment of the invention;

FIGURE 20 shows another embodiment of the invention;

FIGURES 21 to 24 and 26 show another embodiment of the invention;

FIGURE 25 shows details of the embodiments of FIGURES 21 and 24 and 26;

FIGURES 27 to 30 show another embodiment of the invention; and

FIGURES 31 and 32 illustrate alternative shapes for sun blocker components used in the present invention.

Referring now to FIGURE 1, there is shown an oblique view of a fully deployed (i.e. fully unfolded from its launch configuration) spacecraft (or satellite) 1, like the previously described model spacecraft for example, which is represented by a main body 10 which contains six external panels: 11, 12, 13, 14, 15 and 16, a group of

antenna reflectors 20, 21, 22 and 23, and two solar array assemblies, consisting of two solar arrays (one or more solar cell panel) 100 and 101 and their supports 100a and 101a by which they are connected to the main body 10, which are extended northward and southward from the main body out of the north and south panels 11 and 12, respectively. The number of antenna reflectors is driven by the need of the telecommunications application and is a matter of design. In this example, four reflectors are shown and are represented by two deployable large reflectors 20 and 21 mounted on east and west panels 15 and 16, respectively. Two non-deployable reflectors 22 and 23 are mounted on nadir panel 14. While orbiting in a low inclination orbit about Earth, the spacecraft is controlled in such a way that the earth or nadir panel 14 is pointing in the general direction of the center of the Earth, thus allowing the antenna reflectors to perform telecommunications functions with Earth. Opposite to the earth panel 14 is the zenith panel 13.

The solar arrays 100 and 101 may contain multiple panel elements (typically two to eight or more on each side - a four panel-element example is shown in FIGURE 1) or may contain as few as one panel element. However, usually solar arrays comprising multiple solar cell panels are utilized, in order to provide sufficient electrical power for the spacecraft's use. The size and number of the solar cell panels is driven by mission power requirements, and is constrained by, among other factors, the capability of the attitude control subsystem to maintain pointing stability and also by the capability of the thermal control subsystem to manage the heat

dissipated on board the spacecraft. Once the size and number of the panel elements is defined, generally it is desired to maximize the electrical power generated by the solar cells which are mounted on one side of the array panels by facing the cell side of the array toward the Sun as directly and as long and continuously as possible. With spacecraft main body 10 maintaining its earth panel 14 pointing to the Earth continuously, the line between the spacecraft and the Sun will cone around the north-south axis of a spacecraft like the model spacecraft once every orbit, making the Sun appear to circle about the main body 10 as it does so. In order to maintain both solar arrays of a spacecraft like the model spacecraft pointing directly to the Sun they are driven by motor systems which rotate the arrays about the north-south axis, as indicated by the arrow R in FIGURE 5 with respect to the main body 10 at a speed such that the cell side of the array always faces the Sun while the spacecraft orbits the Earth, i.e. the solar arrays rotate about the north-south axis sun synchronously with the Sun to achieve optimum sun exposure for maximum power generation.

Reference is made to FIGURE 2, a top view of prior art spacecraft or satellite 1 of FIGURE 1, wherein the aforesaid seasonal exposures are illustrated. (Parts identical or very similar to those in FIGURE 1 are identically numbered throughout the FIGURES herein and are not all repeated, to reduce redundancy. This applies to all of the following FIGURES that illustrate the same spacecraft or the same parts or components, or ones very similar.) The north panel 11 and the south panel 12

(FIGURES 1 and 2) are maintained oriented parallel to the orbital plane of the satellite, which is co-planar or nearly co-planar with the equatorial plane of the Earth. While the spacecraft is orbiting the Earth, these panels (11 and 12) will not receive daily solar input like the other panels (earth panel 14, zenith panel 13, east panel 15 and west panel 16). Those two panels 11 and 12, however, will be subjected to direct solar heating on a seasonal basis, at incidence angles which will peak at 23.5 degree at the northern summer- and northern winter-solstices respectively, as shown.

FIGURE 3a shows a north-based top view of a spacecraft 1 orbiting Earth at different local times of day and illustrates the constancy with which the nadir panel 14 faces Earth 300 throughout the orbital revolutions. (The solar cell panels are shown edge-on out of the paper.)

FIGURE 3b shows a partial side view of the spacecraft 1 of FIGURES 1 and 2 at midnight, 6 a.m., and noon orbital positions, and also the approximate sun angles at the northern summer and northern winter solstices at midnight and noon.

FIGURES 4a and 4b show the profile of the solar incidence angle on the north and south panels, respectively, such as panels 11 and 12 of spacecraft 1 shown in FIGURE 1, for one calendar year.

It can be seen from FIGURES 4a and b that sunlight is incident on each of the north panel and south panel

for a portion of the calendar year. These periods are nominally 21 March through 21 September for the north panel, and 21 September through 21 March for the south panel. Therefore, the sun ray blocker devices of the current invention perform their shading functions for their respective radiator panels for those periods only.

FIGURE 5 illustrates one preferred embodiment of the current invention, which eliminates or greatly reduces the seasonal solar input on the north and south panels 11 and 12, thus providing more efficient thermal radiators for the spacecraft.

In this present invention embodiment, the sun ray blocker devices (581, 582) comprise two sun blocker components 111 and 112 and mounting, supporting, and deployment mechanisms by means of which the blocker components are integrated with and deployed with the structures and mechanisms that support and cause the solar array to rotate. The radiators on the north and south panels 11 and 12 have dedicated sun ray blocker devices 581 and 582 attached to the north and south array assemblies 100 and 101, respectively, as shown in FIGURE 5. After the thus modified spacecraft 1 has been launched into the operational orbit and its appendages have been fully deployed, the sun blocker components 111 and 112 will achieve their final positions in front of the cell side of the solar arrays with their surfaces more or less parallel to the plane of the solar arrays. The south blocker device 582 is positioned such that during the time between the northern autumnal and northern spring equinoxes, when otherwise there would exist a potential for solar heating

of the south panel 12, the south blocker device 582 will cast a shadow over the south panel 12 thereby eliminating the potential for such solar heating. The north blocker device 581 performs a similar function relative to the north panel 11 during the time between the northern spring and northern autumnal equinoxes. When the solar array assemblies 100 and 101 are maintained directly sun pointed, by virtue of their being rotated, the sun blocker devices will likewise be maintained directly sun pointed and thereby interposed between the Sun and the north and south panels that they shade.

The materials used for the sun blocker components 111 and 112 are selected to minimize the heat transferred from their sun facing surfaces 111a and 112a to their anti-sunward surfaces 111b and 112b. This may be achieved by including insulating material(s) and constructions in the composition of the sun blocker components. For example, sun blocker components may include known thermally insulating materials and assemblies of materials, such as multi-layer insulation (MLI) blankets which utilize layered films of metallized Mylar separated by fabric netting. These materials and constructions are well known in the space industry and have typical heat resistance values of 0.007 to 0.01 Watt/deg.C/sq.in, i.e. 0.0011 to 0.0016 Watt/deg.C/sq.cm. The sun blocker components of the present invention device will generally experience a sizeable temperature difference, for example possibly greater than 100 degree C, between surface 111a and surface 111b and between surface 112a and surface 112b when the satellite is in its normal orientation in the mission orbit, except when the spacecraft is passing through the Earth's shadow.

To obtain the maximum sun blocking effect, the sun blocker components of the sun ray blocker devices 111 and 112 are configured (sized, oriented, and positioned) in such a way that at the summer and winter solstices, when the Sun is about 23.5 degree from the orbit plane, the sun blocker devices will cast shadows that entirely cover the radiator surfaces on their respective thermal radiator surfaces on the spacecraft panels 11 and 12. Accordingly, if the radiator surfaces are rectangular the shadows must be at least as wide as the diagonals of the rectangles.

FIGURES 6a, 6b and 6c show top partial views of a present invention arrangement as the spacecraft orbits Earth and the main body 10 is rotated at the orbital rate so that the earth panel 14 always faces Earth, and the sun blocker components 111 and 112 of the sun blocker devices 681 and 682, respectively, always face the Sun (which is at the left in the FIGURES). These FIGURES are drawn in the inertial frame of reference of the solar array assemblies 100 and 101. Thus, if one were to stand on either of the solar array assemblies 101 and 102 one would see main body 10 rotate one revolution per orbital revolution around the Earth.

FIGURE 7 is a top partial section view showing more details of a present invention spacecraft. In this context the phrase "top view" denotes a view parallel to the planes of the sun blocker components 111 and 112 and also parallel to the orbit plane. Note that in Figure 7 through Figure 15b various examples of embodiments of the present invention are depicted together with generic

partial views of a spacecraft main body and a solar array assembly (labelled 400 and 408, respectively, later in FIGURES 10 and FIGURES 11). Additionally, FIGURES 8 and 9 show alternative embodiment arrangements in top partial section views.

In FIGURE 7, the spacecraft has main body 10, north panel 11, and solar cell panel support 223 with attached solar cell panel 225. In this case, there is a connecting solar array boom-and-yoke 219 and hinges at hinge points 221 and 227. Together this solar cell panel support 223, a solar cell panel 225, a solar array boom-and-yoke 219, and the hinges at hinge points 221 and 227 comprise part of a solar array assembly. The solar array boom-and-yoke 219 fold forwardly against north panel 11 and the solar cell panel support 223 together with the solar cell panel 225 folds down at hinge point 227 in an accordion-like fashion for launching. During launch, ascent, and orbit achievement the solar array assembly is in its folded-closed configuration. After achievement of the mission orbit it is electro-mechanically and/or mechanically deployed (unfolded) to allow the solar cells to be maintained directly sun-pointed. Attached to solar cell panel support 223 is a two-section connecting arm having a short inner portion 209 and an outer portion 207 connected by hinge(s) at hinge point 215. The anti-sunward side 111b of sun blocker component 111 is connected to outer arm portion 207 by hinge(s) at hinge point 203. Optional solar cells 201 are functionally positioned on the sunward surface 111a of the sun blocker component 111. Hinge points 203 and 215 provide for folding of the solar blocker component 111 and its hinged arm 207 against the

solar cell panel 225 in a compact and stiff configuration suitable for launch and subsequent deployment. The electromechanical and/or mechanical designs and methods for deploying (opening) and closing solar array assemblies are commonly used in contemporary spacecraft. The same or similar mechanisms are used to deploy the sun ray blocker devices of the present invention. These mechanisms and methods for deployment and closing are well within the skills of the artisan.

In FIGURE 7, there is an imaginary plane 250 extending off the surface of north panel 11. In its deployed configuration the sun blocker component 111 may touch or extend through this imaginary surface, and consequently may provide additional shading for the earth, west, zenith and east panels as they rotate with respect to the Sun.

FIGURE 8 shows an alternative embodiment where sun ray blocker component 271 does not intersect imaginary plane 250. Further, it has a single connecting arm 205 with hinge points 203 and 217 at opposite ends to form an assembly and is connected directly to the substrate of solar cell panel 225. It may be folded and stowed for launch and deployed or unfolded in orbit in a similar way to the sun ray blocker device in FIGURE 7. In FIGURES 7 and 8, the sun ray blocker devices cast their shadows over the major part of the outer surface of north panel 11 and, in these embodiments, completely shadow that surface during the times when otherwise they would be exposed to the Sun. Further, the solar cells 201 may be

included to produce additional solar power for the spacecraft.

In FIGURE 9, identical parts to FIGURES 7 and 8 are identically numbered. Sun ray blocker component 301 is connected directly to solar cell panel support 223 by hinge(s) at hinge point 309 so as to fold over up-close against solar cell panel 225 in the launch configuration. In this embodiment, sun ray blocker component 301 is not parallel to the solar array, yet still effectively shades north panel 11.

FIGURES 10a, 10b and 10c depict a typical prior art sequence of deployment of a solar array assembly, which is part of the transformation of the spacecraft from its launch configuration to its configuration for normal operations in orbit. For simplification in this document, only one (the north) solar array assembly is shown in the FIGURES. These particular FIGURES show a satellite with a main body 400, and a solar array assembly 408 comprising four solar cell panels, with solar cell surfaces 400a, mounted on solar cell panel supports 408 which are interconnected by hinges at three hinge points 403, 404, and 405 and connected to the main body 400 by a single boom 419 and hinge(s) at hinge points 401 and 402. FIGURE 10a depicts the solar array assembly folded and stowed for launch. FIGURE 10b depicts it in the process of being deployed (unfolded). Figure 10c depicts its fully deployed state. If a multiple-arm boom design is desired by the spacecraft designer, various embodiments can be designed to satisfy

performance requirements using greater numbers of arms and hinge points.

FIGURES 11a, 11b and 11c illustrate the deployment sequence of one possible design embodying the present invention. Components in FIGURES 11a, 11b, and 11c that are identical to components in FIGURES 10a, 10b, and 10c are numbered identically to their identical parts. In addition to the prior art solar array assembly that was previously depicted in FIGURES 10a, 10b and 10c, FIGURES 11a, 11b, and 11c also depict the present invention sun blocker component 411 connected to the solar array boom 419 by an arm 430 and hinges at hinge points 406 and 407. Alternatively, by design the sun blocker component 411 could be hingedly attached via the arm 430 to a convenient different location on the solar array assembly. FIGURE 11a depicts the solar array assembly and the sun ray blocker device folded and stowed for launch, FIGURE 11b shows them partially deployed (unfolded). FIGURE 11c depicts their fully deployed state. FIGURES 12a and 12b show sun blocker components which are not parallel to the plane containing the solar cell panels yet which still provide proper shading of the north or south panel. Components in FIGURES 12a and 12b and subsequent figures that are identical to components that appear in previous figures are numbered identically with their corresponding or very similar components or are left un-numbered to avoid unnecessary repetition. Alternatively, by design the sun blocker component 111 could be hingedly attached via the arm 430 to a convenient different location on the solar array assembly.

FIGURE 13 depicts yet another alternative embodiment of the present invention. The sun blocker component 511 is connected to the solar array boom 219 by hinge(s) at hinge point 507 for its stowing folded and subsequent deployment.

FIGURES 14a and b show an arrangement similar to that in FIGURE 13, with identical parts identically numbered, however, more hinges at hinge points 606 and 607 are used with sun blocker component 611 as required by design for folding the sun blocker components prior to deployment.

FIGURE 14b represents a partial view of the anti-sun side of the spacecraft looking toward the Sun (i.e. a side view relative to the top view shown in Figure 14a).

FIGURES 15a and 15b show one embodiment in which sun blocker component 811 utilizes separate active motors 306 and 307 which are used to actively deploy and/or retract the sun ray blocker device. This arrangement allows satellite operators to use deployment motors that are separate from the solar panel deployment motors so as to permit them to retract the sun blocker devices to prevent their interference, if any, in satellite operations such as in the use of propulsion systems during spacecraft performance of station keeping or attitude control maneuvers.

In some spacecraft designs the required size (dimensions and/or area) of a sun blocker device in its fully deployed configuration may exceed the constraints of its "launch envelope" i.e. the constraints of the

maximum-allowable space allocated to the sun ray blocker device in the launch configuration of the spacecraft when the solar array and the sun ray blocker device are in their launch configuration. Therefore, for compatibility with the constraints of the size of the corresponding launch envelope it may be necessary for the sun blocker component of the sun ray blocker device to comprise several (i.e. more than one) pieces, instead of being one single integral piece, which are folded together in the launch configuration and are subsequently deployed (unfolded) in orbit to form effectively one continuous sun blocker component. FIGURES 16a, 16b and 16c, and FIGURES 17a, 17b and 17c, respectively depict two examples from the many possible designs for sun blocker components which fold and deploy. Parts a, b, and c of the FIGURES 16 and 17 show each of the two designs in a view from the front (Sun) direction with the sun blocker device fully deployed, and two views from a direction orthogonal to both the front view and the top view directions with the sun blocker device folded and deployed, respectively. (As defined earlier herein the phrase "top view" denotes a view that is simultaneously parallel to the plane of the sun blocker components 921 or 951 and the orbit plane.) This allows the sun ray blocker device to increase its dimensions using hinge(s) and/or strut(s) and/or flap(s) etc. at hinge points 925 and 927 or a slide-out design. Referring collectively to all FIGURES 16, sun blocker component 921 has a center section 923 with hinge(s) and/or strut(s) and/or flap(s) etc. at hinge points 925 and 927 and outer, swing up sections 929 and 931 which may be designed to deploy (swing up) automatically. In all FIGURES 17, sun blocker component 951 has main section 953

with slide-out extensions 955 and 957 that may be designed to deploy (slide out) automatically. (Automatic hinging and automatic sliding or telescoping is well within the purview of the artisan in the spacecraft industry and need not be further elaborated upon herein.)

The embodiments of the present invention device illustrated in FIGURES 18 through FIGURE 30 are as generally applicable as the other embodiments described herein. However, they also function efficiently in cases where a sun blocker device cannot be attached to an axle located near the center (1811, 2123, 2722) of an associated thermal radiator surface (1804, 2121, 2721).

One such case is that in which the axis of rotation (1803, 2131, 2701) of a solar array assembly extends outward from the associated thermal radiator surface (1804, 2121, 2721) at a location that is significantly offset from the center (1811, 2123, 2722) of the thermal radiator surface. In that case, designs with an attachment arm of fixed length between the sun blocker component and the solar array axis could be unsuitable, because the motion of the sun blocker component about the center of the thermal radiator surface would be eccentric.

Another such case is that in which there are stay-out zones inboard of the periphery of the associated thermal radiator surface - through which objects such as a supporting boom (for example for a sun blocker component) are not allowed to pass. This could be the case, for example, when certain attitude- or orbit-control

thrusters (1810) are also located on the main-body panel of the spacecraft upon which the thermal radiator surface (1804) is located.

The arrangements illustrated in FIGURES 18 through FIGURE 30 may be employed to overcome these constraints, whilst still maintaining a sun blocker component at a substantially uniform distance from the center of the associated thermal radiator surface and achieving compact static and swept volumes of a sun blocker device. Selection between the embodiments shown in FIGURES 18-30 for any particular application may involve trade-offs between many additional performance-requirements of the spacecraft-system, including for example: mass, strength, stiffness, flatness, circularity, simplicity, and reliability.

Figure 18 shows an embodiment of a sun blocker device in which the sun blocker component 1800 is mounted on a carriage 1801 with wheel-sets or bearing-sets 1808 and 1830 by means of an attachment arm 1805 in which the carriage may be driven around a closed guide 1802 in at least one of the directions of the arrows 1832, the carriage 1801 being attached to the guide 1802 by rolling or sliding means that also react against and thereby limit rotations of the carriage 1801 (and thereby the sun blocker component) about axes passing through the points of contact of the carriage and the guide. In one of many potential embodiments, for example, this may be achieved using wheel or bearing sets 1808, 1830 that are adequately spaced both along-track and cross-track on both sides of the guide 1902, and which are also cambered at an adequate angle to the plane of the baseplate.

Attached to the carriage is at least one generally-radial boom or strut 1805, an outer end of which is attached to

the sun blocker component at hinge point 1812 and an inner

end of which is attached to the carriage 1801 at hinge-point 1813, and the carriage is rollingly or slidingly mounted on the guide 1802 by the wheels or bearings 1808 and 1830. At least one of these wheels or bearings 1830 is provided with a motor to rotate the wheel in at least one of the directions of the arrows 1831 and thereby drive the carriage along the guide 1802, for example by friction, or by the engagement of a toothed wheel or a worm-drive in a rack. Electrical power may be supplied to the motor, via brushes for example. The attachment arm 1805 and the sun blocker component 1800 can be folded at the hinge points 1812 and 1813 to achieve a stowed configuration of the sun blocker device for launch, during which the folded device may be temporarily caged securely for proper management of launch-induced dynamic environments and loads. The sun blocker device may be further folded for launch as illustrated in FIGURES 16 and 17. Following launch the attachment arm 1805 and the sun blocker component 1800 can be deployed for subsequent operation in orbit, including sun-tracking travel around guide 1802. It will be appreciated that the guide 1802 need not be circular as shown in FIGURE 18, but in the case of a significantly rectangular thermal radiator surface, for example, the guide could be elliptical and in either case may be diverted to avoid obstacles mounted on the spacecraft.

Alternatively, as illustrated in FIGURE 19 the attachment arm 1905 could be mounted on a solid rotatable wheel instead of on a carriage and guide, for example by an intermediate structure 1901, shown comprising elements 1906 and 1907. In the embodiment illustrated in FIGURE

19 the wheel is a ring or annulus 1902 floating in circumferentially located bearing-sets 1903 and

controlled and driven by a motor 1930 mounted on the baseplate under 1804. The outer end of attachment arm 1905 is attached to the sun blocker component at hinge point 1812, and the inner end to the intermediate structure 1901 at hinge-point 1913. Arrows 1931 and 1932 indicate rotation of the motor 1930 and resulting rotation of the sun blocker component, respectively.

In a similar alternative embodiment, illustrated in FIGURE 20, a carriage 2001, similar to that provided in the embodiment illustrated in FIGURE 18, is provided; but in this embodiment the carriage 2001 is driven around a closed guide 2002 in at least one of the directions of the arrows 2032 not by a motorized wheel, but by external motive means such as an endless belt 2003, chain, or cable attached to the carriage 2001, the belt for example being driven by a motor 2030 that is mounted to the baseplate under 1804 and which engages the belt 2003, chain, or cord, and rotates in at least one of the directions of the arrows 2031. A tensioning device 2040 is also provided to engage the belt 2003 and tension the belt while not impeding the passage of the carriage around the guide 2002, for example by exerting a force on the belt in the direction of arrow 2042. Again, as in the embodiment illustrated in FIGURE 18 the guide 2002 need not be circular.

FIGURES 21 through 30 illustrate embodiments in which a sun blocker component is mounted on the spacecraft via an attachment arm from an axis 2131, 2701 that is offset from the center 2123, 2722 of an associated thermal radiator surface. The axis 2131, 2701 could be concentric with or identical to the axis of rotation of a solar array assembly.

FIGURES 21 through 26 illustrate an alternative embodiment in which a sun blocker component 2100 is attached to an axle at axis 2131. The axle may be concentric with or identical to an axle of a solar array assembly. The sun blocker component is attached to the axle by an articulated attachment arm 2130 that includes three articulated portions 2132, 2134, 2137.

The inner end of the inner-portion 2132 is fixed radially to the axle at axis 2131. The middle-portion 2134 is pivoted at its inner end to inner-portion 2132 at pivot-point 2133, and the outer end of middle-portion 2134 is pivoted to the inner end of outer-portion 2137 at pivot point 2135. At its outer end the outer-portion is attached to the sun blocker component at hinge point 2138 and near its inner end the outer-portion is hinged at hinge point 2136 to allow folding and stowing for launch followed by deployment in orbit.

As depicted in FIGURE 21 through FIGURE 26 the inner-portion 2132 and the outer portion 2137 of the attachment arm 2130 turn anti-clockwise at the same rate, the outer-portion 2137 carrying the sun blocker component with it, whereas the middle portion 2134 rotates clockwise at the same rate.

The length of the inner-portion 2132 is approximately equal to the offset of the axis of rotation 2131 from the center 2123 of the thermal radiator surface. In principle the length of the middle-portion 2134 may be longer or shorter than the length of the inner portion 2132. However, in the case that the axis of rotation 2131 is occupied by an obstruction such as the axle of a solar array assembly then the middle-portion 2134 must be shorter than the inner-portion 2132 for clearance of the solar array axle at axis 2131, as can be seen in FIGURE 23 in which the attachment arm 2130 is approaching its closest to the axle at axis 2131.

By articulating the articulated portions through rotation of the arm 2130 about the axis of rotation 2131 the sun blocker component can be maintained at a substantially constant distance from the center of the associated thermal radiator surface 2121, to describe a substantially circular path 2140 around the spacecraft. It will be evident that in the case of a thermal radiator surface that is significantly far from being radially symmetric the length of the articulated portions of arm 2130 could be adapted to achieve a wide range of desired paths around the thermal radiator surface.

As shown in FIGURE 21, the articulated portions 2132, 2134, 2137 are arranged to the full reach of attachment arm 2130 in a straight line when the sun blocker component is passing a side of the thermal radiator surface furthest from the axis of rotation 2131. As shown in FIGURES 22 and 23 the attachment arm 2130 has an effective length equal to the sum of the lengths of an outer 2137 and an inner 2132 articulated portion when the sun blocker component 2100 is at an intermediate distance from the axis of rotation 2131. As shown in FIGURE 23, the effective length of the attachment arm 2130 is at its minimum when the sun blocker component is at its closest to the axis of rotation 2131, at which point its length is equal to the sum of the lengths of the inner 2132 and outer 2137 portions less twice the length of the middle-portion.

The inner articulated portion 2132 of the attachment arm 2130 rotates about axis 2131. The means of attachment of the inner articulated portion 2132 may be

independent of a solar array axle along axis 2131, the inner portion 2132 then being mounted to a concentric tubular axis around a central solar array axle. Alternatively, the inner articulated portion may be fixed solidly to a solar array axle along axis 2131.

In the illustrated embodiment, for a geostationary spacecraft for example the inner and outer articulated portions rotate anti-clockwise at one revolution per day, and the middle articulated portion rotates clockwise at one revolution per day. This rotational relationship may be achieved by diverse means, such as: separate motorized pivots at pivot points 2131, 2133, and 2135; or by a system of belt-linked pulley wheels at pivot points 2131, 2133 and 2135, driven by a single motor or by an axle along axis 2131.

The articulated portions 2132, 2134, and 2137 may be sprung together, so that in a failure mode the attachment arm 2130 automatically extends to its greatest length. In that case, any failed pivot points can be made to fail free, for example using commandable frangible-links in the associated pulley wheels or drive motor, allowing spring-driven extension of the arm 2130.

FIGURES 25 and 26 illustrate a means of articulating the articulated portions 2132, 2134, 2137 with respect to each other, utilizing a driving force from a solar array axle at axis of rotation 2131. A cylinder 2501 is provided, mounted co-axial with the solar array axle at axis 2131 but fixed to a base panel 2121. The inner articulated member 2132 is fixed to the solar array boom

at axis 2131 so that the inner articulated member 2132 rotates at the same rate as the solar array boom at axis 2131. A middle articulated portion 2134 is pivotedly fixed to an outer end of the inner articulated portion 2132 at a pivot point 2133 and a pulley wheel 2502 of the same diameter as cylinder 2501 is fixed to an inner end of the middle articulated portion 2134. A toothed belt 2506 is looped around the cylinder 2501 and the pulley wheel 2502 so that as the solar array shaft 2131 and the inner articulated portion 2132 rotate anti-clockwise in a direction of the arrow 2507, the toothed belt 2506 causes the pulley wheel 2502 and the middle articulated portion 2134 to counter-rotate at the same rate in the direction of arrow 2508.

As shown in FIGURE 26, a second equal sized pulley wheel 2601 is fixed to an outer end of the inner articulated portion 2132 on a side of the inner articulated portion 2132 opposite to that on which the cylinder 2501 is fixed to base panel 2121, and a third equal sized pulley wheel 2602 is fixed to an inner end of an outer articulated portion, such that the third pulley wheel 2602 and the outer articulated portion 2137 are together pivotedly attached to the outer end of the middle articulated portion 2134. A second toothed belt 2603 loops around the second pulley wheel 2601 and the third pulley wheel 2602 so that as the middle articulated portion 2134 rotates in the direction of arrow 2508 the toothed belt 2603 causes the outer articulated portion 2137 and the third pulley wheel 2602 to counter-rotate at the same rate in the direction of arrow 2604. Thus, the outer articulated portion 2137 rotates in the same sense

as the inner articulated portion 2132 whereas the middle articulated portion 2134 counter-rotates.

In a further embodiment illustrated in FIGURES 27 through 30, a sun blocker component 2700 is attached to an axle 2701 of a solar cell array by means of a scissors attachment arm 2730. The scissors arm comprises a first articulated arm 2704, 2708 and a second articulated arm 2705, 2709, comprising inner articulated portions 2704, 2705 and outer articulated portions 2708, 2709 respectively. The inner articulated portions are connected by hinges at hinge points 2702, 2703 to the solar array boom 2701 respectively and the outer ends of the outer articulated portions 2708, 2709 are connected by hinges 2710, 2711 to the sun blocker component 2700 such that when the articulated arms are extended to the full length they are still not parallel to avoid their locking up. A lanyard 2712 is located in between the articulated arms 2704, 2708 and 2705, 2709 and extends between the sun blocker component 2700 and the solar array axle 2701. The inner articulated portions 2704, 2705 and the outer articulated portions 2708, 2709 are sprung at hinge points 2706 and 2707 so as to automatically extend the articulated arms 2704, 2708 and 2705, 2709 to their full extent as limited by the lanyard control. The articulated arms 2704, 2708 and 2705, 2709 thereby form a parallelogram, the shape of which may be controlled by retracting or deploying the lanyard 2712. Alternatively the shape of the parallelogram could be controlled by motorized hinges, or alternatively by a retractable and deployable lanyard between hinge points 2706 and 2707 with sprung hinges 2702, 2703, 2710 and 2711 instead of

at 2706 and 2707. In the embodiment of the invention illustrated in FIGURES 21 through 24, the distance of the sun blocker component 2700 from the solar array boom 2701 can be varied as the sun blocker component 2700 rotates about the solar array boom 2701 to maintain the sun blocker component at a constant distance from the spacecraft as illustrated by the path 2712.

The embodiments described in FIGURES 18 through 20 have the advantage that the attachment arm does not obscure thrusters 1810 present on the face of the spacecraft, that the sun blocker component shades.

A sun blocker component 3100, 3200 is not necessarily rectangular in shape. As shown in FIGURE 31, the sun blocker component 3100 has trapezoidal first-and second- extensions 3101, 3102 hingedly attached to a main body 3103 of a sun blocker component 3100. The first extension 3101 is extended by unfolding the extension through rotation in the direction of the arrows 3104 and the second extension is extended by unfolding the extension through rotation in the direction of the arrows 3105 from a position flat against the main body 3103.

As shown in FIGURE 32 a rectangular main body 3202 of the sun blocker component 3200 may have substantially triangular extensions 3201, 3202 which may be extended and retracted from the main body by sliding translation of the extension 3201 in the direction of double-handed arrow 3204 and unfolding the extension 3202 in the direction of arrows 3205.

FIGURES 5 to 9, 11 to 24, and 26 to 32 illustrate sun blocker components each of which includes a region of an anti-sun-facing surface adapted to lie, in an operational configuration, substantially in a plane. FIGURES 5 to 8, 11, 12b, 15, 18 to 24, and 26 to 32 illustrate sun blocker components each of which includes a region of an anti-sun-facing surface adapted to face, in an operational configuration, at an angle away from an associated thermal radiator surface.

The descriptions of designs for the structural support and the deployment of sun ray blocker devices written herein are examples from thousands of possible structural support and deployment designs which can be used for this purpose and are within the scope of the present invention.

This paragraph describes an example to demonstrate the geometrical approach to calculating the dimensions of a sun blocker component for providing total shadow coverage to a quasi-rectangular shaped radiator surface. The example used is that of a radiator surface on a north or south panel of a geostationary spacecraft, like the previously defined "model" spacecraft for example, at the summer or winter solstice, when the incidence angle of the Sun's rays (measured from the plane of the benefited thermal radiator surface) is at a maximum, using a quasi-rectangular (for this simple illustration at least) shaped sun blocker component whose plane is perpendicular to the plane of the associated thermal radiator surface (referring to FIGURE 12a, angle 501 is then 90 degree). For this example, take the north or south radiator-surface of the spacecraft to be rectangular, of length and width A and B, respectively. Then, the length and width dimensions, L and W, respectively, of the sun-exposed surface of the fully-deployed sun blocker component (which is shown in FIGURES 16a and 17a) should be as follows: L is greater than or equal to $\sqrt{(A^2+B^2)}$, and W is greater than or equal to 0.435x $\sqrt{(A^2+B^2)}$ based on the corresponding orbit-sun angle of 23.5 degree. However, if only a portion of the surface area on the north or south panel needs to be shadowed, i.e. high heat-dissipating

equipments were to be mounted in certain localized areas of the north or south panel, the sun ray blocker device can be tailored to shade only those areas and may accordingly be smaller. In addition, if a sun blocker component whose plane is not perpendicular to the plane of the associated thermal radiator surface were to be selected by the spacecraft designer, the minimum value of the width, W, may be greater or less than $0.435x \sqrt{(A^2+B^2)}$ depending on the size of angle 501 in FIGURE 12a. If angle 501 is greater than 90 degree, W may be greater than $0.435x\sqrt{(A^2+B^2)}$; if it is less than 90 degree, W may be less than $0.435x \sqrt{(A^2+B^2)}$. If additional shading to the other four panels, earth, zenith, east and west panels, is desired, the width (W) of the sun blocker component can be increased to extend past the imaginary plane 250 toward the center of the satellite as shown in FIGURE 7.

Thus, by the foregoing descriptions contained herein it can be seen that by virtue of the present invention losses in the efficiency of the cooling of the thermal radiator panels of a spacecraft caused by solar heating can be eliminated or minimized via various sun blocking arrangements.

Obviously, numerous modifications to and variations on the present invention are possible in light of the above teachings. For example, as a practical matter, a designer might counterweight or counterbalance the rotating axles or arms to overcome the weight imbalance caused by sun ray blocker devices of the present invention without exceeding the scope of the present

invention. It is therefore understood that within the scope of the appended claims, the invention may be practised otherwise than as specifically described herein.